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"THE DESIGN, CONSTRUCTION AND FLIGHT TESTING OF A LARGE LIQUID PROPELLANT MISSILE"

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"THE DESIGN, CONSTRUCTION AND FLIGHT TESTING OF A LARGE LIQUID PROPELLANT MISSILE"

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THE DESIGN, CONSTRUCTION AND FLIGHT TESTING OF A LARGE LIQUID PROPELLANT MISSILE

ABSTRACT

This paper describes the design, construction, and testing of a liquid fueled rocket. Design calculations for the engine and propulsion systems were performed and an engine was constructed to those specifications. A static test stand was constructed and instrumented to evaluate engine performance during full duration static firings.

The engine and propulsion system proved successful and were then incorporated into the flight vehicle. The remainder of the paper describes the design, construction and testing of all flight hardware including an on-board camera system and recovery system.

Two unsuccessful attempts to launch the missile were made on 11 May 1976 at White Sands Missile Range in New Mexico. The reasons for the failure were corrected and the missile was successfully launched on 17 May 1976.

TABLE OF CONTENTS

The work recorded in the following pages is of no great significance to the fields of rocketry or propulsion. Rather, this project was a tool which afforded the experimenter the opportunity and facilities with which to learn the maximum amount of practical engineering possible over the course of eight months. The field of rocketry was chosen because it alone encompasses so many other disciplines (i.e., chemistry, physics, thermodynamics, construction techniques, electronics, etc.) and because no other field so dramatically and unforgivingly demonstrates the ability of the designer. The goal of the project was to design and fly a missile, but its purpose was to teach the experimenter 1) enough engineering to design a workable piece of hardware and 2) how to take a design and turn it into hardware. This project has accomplished both its goals and its purpose.

The initial step in the design of a rocket is the assignment of certain overall vehicle parameters. Once established, these parameters dictate the size, shape, capabilities, and limitations of the system. These parameters also dictate the required performance and limitations for all subsystems. The propulsion requirements for the intended vehicle were first defined and a propulsion unit designed accordingly.

The thrust was set at 1000 pounds because this value promised good performance and size without complicating the construction and feed problems. Once a value for thrust was

assigned, the engine could be designed and burn time, propellants and fuel system type would determine overall size and weight of the vehicle.

Burn duration was set at 15 seconds to keep the total loaded vehicle weight down and thereby assure a high takeoff thrust to weight ratio and thus high takeoff velocity. Lack of a high initial velocity would require either some sort of active guidance or complex launch equipment--both of which were beyond the realm of an eight month project.

Liquid oxygen and kerosene were chosen as propellants for their high specific impulse, ease and safety of handling and low cost. While the cryogenic nature of liquid oxygen presented several unique problems, they were far less than those presented by red fuming nitric acid or similar oxidizers.

A pressure feed system was decided upon for simplicity and efficiency. Vehicle size and the problems involved in the manufacture of pumps ruled them out as a feed system. Nitrogen was used as the pressurizing gas because of its inertness, and ease of procurement. Helium was not used because of the leak problems inherent to high pressure systems using it as a pressurizing gas.

In the area of guidance, it was decided that the rocket should be ballistic and only fin stabilized. The design and manufacture of a workable guidance system would undoubtedly have taken more than a year by itself, requiring several rocket flights to test and prove the system.

To summarize then the initial vehicle parameters were:

Thrust	1000 pounds
Burn time	15 sec.
Oxidizer	Liquid oxygen
Fuel	Kerosene
Feed system	3000 psi nitrogen
Guidance	Fin stabilized

Therefore, within these limits, the engine was designed. However, before a complete design could be generated, other decisions directly related to the engine had to be made. The chamber pressure was arbitrarily assigned a value of 300 psia. Lox-kerosene combustion gives good performance at this pressure and low chamber pressure would reduce weight and nitrogen feed requirements. Cooling of the chamber was the next problem to be faced. Internal combustion temperatures for lox-kerosene at 300 psi run on the order of 6000°F. With a 15 second burn the engine and nozzle had to be cooled in order to survive.

The two primary methods used for cooling liquid rocket motors are film and regenerative. Film cooling[1] was attractive due to its simplicity and uncomplicated construction, but regenerative[2] cooling was more efficient and controllable. In this engine both were to be incorporated to ensure there would not be an engine burn-through due to loss of cooling.

The rest of the engine dimensions were calculated as follows: The thrust coefficient, C_F, is defined as

$$C_F = \sqrt{\left(\frac{2K^2}{K-1} \quad \frac{2}{K+1}\right)^{(K+1)/K-1} \left(1 - \frac{p_2}{p_1}\right)^{(K-1)/K}} + \frac{p_2 - p_3}{p_1} \frac{A_2}{A_t}$$

where K = the specific heat ratio, p_1 is the chamber pressure, p_2 is the atmospheric pressure and p_3 is the nozzle exit pressure. The specific heat ratio can be interpolated from tables[3] to be 1.24 for the condition present, p_1 is 300 psia and since the nozzle exit pressure is equal to atmospheric pressure ($p_2 = p_3$) the last term of the equation is equal to zero. A_2/A_t is the area ratio of the exit to the throat.

$$C_F = 1.407$$

The area of the throat can be calculated from the equation

$$A_t = \frac{F}{C_F p_c}$$

where A_t = area of the throat, F = thrust, and p_c is equal to the absolute chamber pressure.

$$A_t = 2.36 \text{ in}^2$$

This ideal throat area is then corrected for throat shrinkage in the hot fire condition and boundary layer effect

$$A_{real} = A_t \times (.987) = 2.33 \text{ in}^2$$

The nozzle throat diameter is therefore

$$D_t = \sqrt{\frac{4A_t}{\pi}} = 1.724 \text{ inches}$$

The nozzle exit area can be found by multiplying the expansion ratio, ε, by the area of the throat. ε can be obtained from tables[4] and is equal to 4 in this case.

The area of the exit nozzle therefore equals

$$A_E = A_t \, \varepsilon = 9.40 \text{ in}^2$$

and the diameter of the exit equals 3.45 inches.

Converging and diverging half angles of 45^o and 11^o respectively were chosen and the finished nozzle length was calculated.

A cylindrical combustion chamber was desired and a diameter of 4.0 inches was chosen to meet size requirement. A characteristic chamber length, L*, of approximately 40 inches is common for lox-kerosene systems and so was used in this case. Since L* is defined as the ratio of chamber volume to throat area the required chamber volume is

$$V_c = L^* \, A_t = 96 \text{ in}^3$$

With an inside diameter of 4.0 inches this gives a chamber length of 6.625 inches.

A drawing of the engine and cooling jacket can be seen in Figure 1.

Ideal exhaust velocity is found from the ideal specific impulse (I_{sp}) which in this case is approximately 252 sec.

$$V_2 = I_{sp} \, g = 8118 \text{ ft/sec}$$

The characteristic exhaust velocity, C*, is equal to

$$C^* = \frac{\underline{Ispg}}{C_F} = \frac{V_2}{C_F} = 5762 \text{ ft/sec}$$

C* is then corrected for losses in the chamber, leaving

$$C^*_{experimental} = C^*_{ideal} \times .9 = 5185 \text{ ft/sec}$$

The propellant flow rates and injector specifications were the next to be determined. A multiple hole triplet impinging injector design was decided upon for two reasons: 1) this type of injector gives excellent results by ensuring good mixing and a hot burn; 2) this design allows the oxidizer holes to emerge normal to the injector face. This ensures that if the fuel runs out before the oxidizer, lox will not be sprayed onto a hot chamber wall and burn through. A drawing of the injector can be seen in Figure 2. The propellant weight flow, $\dot{W}$, can be found from

$$\frac{P_c \; A_t \; g}{\dot{W}} = C^*. \quad \text{Therefore:}$$

$$\dot{W} = \frac{P_c \; A_t \; g}{5185} = 4.253 \text{ pounds/sec}$$

If r is the mixture ratio and is assigned a value of 2.00, the fuel flow rate $\dot{W}_f$ is equal to

$$\dot{W}_f = \frac{\dot{W}}{r+1} = \frac{4.253}{3.00} = 1.417 \text{ lbs/sec}$$

The oxidizer flow rate is therefore

$$\dot{W}_o = \dot{W} - \dot{W}_f = 2.835 \text{ lbs/sec}$$

The injection flow volumes are

$$Q_o = \dot{W}_o/\rho_o = 68.90 \text{ in}^3\text{/sec}$$

$$Q_f = \dot{W}_f/\rho_f = 48.39 \text{ in}^3\text{/sec}$$

The injector hole areas can be calculated from the mass flow rate. A pressure drop of 80 psi across the injector is assigned to both the fuel and oxidizer feeds, and a discharge

coefficient, Cd, of .70 is assumed. The total oxidizer injection hole area is

$$\Sigma A_o = \frac{W_o}{Cd\sqrt{2g\Delta p\rho_o}} = 0.08616\ \text{in}^2$$

For 24 oxygen holes in the injector the individual hole area is

$$\frac{0.08616}{24} = 0.0036$$

which gives a hole diameter of 0.0676 inches.

The total fuel injection hole area is

$$\Sigma A_f = \frac{W_f}{Cd\sqrt{2g\Delta p\rho_t}} = 0.05086\ \text{in}^2$$

Film cooling is employed at the injector face by injecting about 10% of the total fuel flow along the chamber wall. This is accomplished by drilling a ring of holes around the injector face very close to the wall of the motor. This fuel is injected outside the mixing zone and thus is never directly combined with the oxidizer. This circular zone of extra fuel injection helps cool the chamber down to a second row of cooling holes drilled just above the converging section of the nozzle. This lower row is not included in the total fuel flow. Extra fuel space is allotted in the kerosene tank for the flow through these lower cooling holes. There are 24 of these holes, each 0.013 inch in diameter.

Since the total fuel hole area is 0.05086 in^2 the cooling hole area for the injector is 0.00509 in^2, 32 such holes give

an individual hole area of 0.000159 in^2 and a diameter of 0.014 inches. The remaining 90% of the fuel hole area is divided into 48 holes, giving an individual hole area of 0.00095 and a hole diameter of 0.0348 inches.

Injection velocities can be calculated from the following equations (assuming there is no jet contraction).

$$V_f = Cd \; \frac{2g\Delta p}{\rho_f} = 84.76 \text{ ft/sec}$$

$$V_o = Cd \; \frac{2g\Delta p}{\rho_o} = 71.50 \text{ ft/sec}$$

These velocities are high enough to give good injection and mixing. They also justify the original assumption of a 75-80 psi injector Δp.

Since the injector is of a triplet impingement design, the resulting momentum of the mixed propellant will be in an axial direction without adjusting the injection hole angles. The rest of the injector design (feed connection points, manifolds, seals, etc.) was done graphically with an emphasis on ease of construction. Figure 2 shows the completed injector design.

As was stated earlier, the motor was to be both regenerative and film cooled. Film cooling was accomplished by adding the extra fuel injection holes near the converging nozzle and on the injector face. The regenerative system was produced by building a jacket around the motor itself. In such a system the velocity of the fuel flowing around the motor is

critical. If the flow velocity is too slow, boiling of the coolant will result and cause loss of heat transfer and a burn through. Too high a flow velocity causes excessively high pressure drops through the jacket.

The flow velocity around the chamber and nozzle exit were sec at 20 ft/sec which promised good cooling without exhorbitant pressure drops. Since the heat transfer rate is much higher at the nozzle throat, the velocity at this point was increased to 40 ft/sec. With the flow velocities assigned, the dimensions of the coolant passage are determined by the coolant density and the cross sectional area of the passage.

$$A = \frac{\dot{W}_f}{\rho_f V}$$

The original jacket design called for an annular coolant passage. However, this design proved impossible to construct. To ensure the correct velocities of coolant at any given point along the chamber wall, tolerances had to be kept within ±.002 inches. At the throat the passage between the inner and outer walls was only 0.010 inches. Thermal shock, warpage expansion and contraction during engine start up might close off such a narrow passage causing a burn through. For this reason the coolant passage was redesigned as a helical rather than annular one. This allowed the jacket design to remain the same, but by forcing the coolant through a 1.5 inch wide passage formed by wrapping and silver soldering a coil of

copper wire to the engine, the distance between the inner and outer walls could then be increased to almost 0.080". The velocities at different points along the wall could be regulated by widening or narrowing the distance between consecutive coils. From the preceding equation the cross sectional area of the coils can be calculated. At the throat where the velocity of the coolant should be 40 ft/sec

$$A = \frac{\dot{W}f}{\rho_f V} = 0.0828 \text{ in}^2$$

With a 0.080 passage between the jacket walls, the coil width should be approximately 1.035 inches wide. For a velocity of 20 ft/sec around the chamber and nozzle exit the passage should be twice as wide or approximately 2.060 inches.

The pressure drop through the cooling coil can be approximated by summing the loss through each wind

$$\Delta p = \frac{f\ell}{dh} \frac{\frac{W_f}{A_c}}{2g\rho_f}$$

where A_c is the cross sectional area of the coil, f is the friction coefficient, d_h is the hydraulic diameter, and ℓ is the coil length. For a rectangular passage the hydraulic diameter can be assumed to be two times the thickness of the passage. Calculation of Δp for the motor gave a value of 137.4 psi. This value is high but understandable for the dimensions of the coil.

Heat transfer values can be approximated and are useful here as an indication of engine performance. Neglecting the

film cooling, the heat problem can be solved as follows. The average diameter of the nozzle-chamber combination is 3 inches and the entire motor length is 13 inches. Therefore, the total surface area is approximately 122.5 in^2. Assuming an average heat flow of 1.5 BTU/in^2 sec the total heat transfer is

$$Q_{rej} = (1.5)\ (122.5) = 183.8 \text{ BTU/sec}$$

But

$$Q_{rej} = \dot{W}_f\ C_p\ \Delta T_{rise}$$

where C_p is the average specific heat of kerosene at the temperature in question, and in this case, is equal to 0.54 BTU/lb-^oR. Therefore

$$\frac{Q_{rej}}{\dot{W}_f\ C_p} = = \Delta T_{rise} = 259.4^oF$$

This temperature plus the ambient temperature of the fuel will give the injection temperature.

$$T_{inj} = T_{rise} + 70^oF = 329.4^oF$$

This T_{inj} is not excessive and promises good performance. The wall temperature at the throat can now be calculated. The film coefficient of the coolant can be calculated to be

$$h_c = .002\ \rho_f\ V_f\ C_p = 0.0153 \text{ BTU/in}^2 \text{ sec }^oR$$

At the throat the heat flow can be assumed to be approximately 2.25 BTU/in^2 sec. Therefore

$$\Delta T_f = \frac{2.25}{h_c} = 147^oF$$

The temperature of the coolant after traveling from the entrance to the jacket to the throat (approximately 1/3 of the total cooling coil length) is equal to:

$$T_c = 70 + .333\ (259.4) = 156^{o}F$$

Temperature of the wall on the coolant side equals

$$T_{wc} = T_c + \Delta T_f = 303^{o}F$$

ΔT across the wall is equal to

$$\Delta T_w = \frac{2.25\ (t_w)}{k_w}$$

where T_w is the wall thickness and k_w for mild steel can be calculated to be 0.000509 BTU/in^2 hr ^{o}F/in. Therefore

$$\Delta T_w = \frac{(2.25)(.062)}{.000509} = 274^{o}F$$

Therefore, the temperature of the wall on the gas side is

$$T_{wg} = 274 + 303 = 577^{o}F$$

This temperature is well within the acceptable limits for the materials being used and indicates that the designed cooling passage will sufficiently cool the motor during operation.

The designing having been completed, the engine was then constructed. A solid steel mandrel with its contours machined to the calculated inside dimensions of the motor was made. This mandrel was used to spin the actual motor shell. A mild steel tube was rolled and welded, then slipped over the mandrel and spun to the final shape. The mandrel was then removed and the ends of the motor shell were machined to their final length. Two such motor shells were made.

Flanges were then heliarced to both ends of the motor. The upper one would hold the injector in place and the lower one would form the bottom of the fuel feed manifold for the jacket. The coolant passages were formed by silver soldering a coil made out of copper wire around the motor. As per calculation, the coolant passage width varied from 2 inches down to one to get the different velocities needed. This copper coil was then machined to a thickness of 0.080 inches.

The next step was production of the jacket itself. At first it was thought that the outer jacket could be made out of aluminum and could be bolted around the motor. This way, if the engine shell burned through during the static firings, a new one could be installed without remaking the jacket. This idea proved unfeasible due to the sealing problems encountered. It was then decided to go to a welded steel jacket.

The jacket was manufactured in two pieces and assembled out of three. A thin walled tube was machined to fit over the chamber portion of the engine and then a nozzle section was machined to fit outside the nozzle. This nozzle jacket was split in half and then heliarced in place around the nozzle and to the chamber jacket. Two stiffening bands were added around the chamber jacket to ensure that the seamed pipe (out of which the jacket was made) did not burst.

All joints of the jacket were heliarced except one. The junction of the jacket to the upper engine flange was

silver soldered together. The upper flange of the engine was only 0.250 inches thick and calculations showed this to be a little on the thin side. Welding to the jacket would have caused warpage, and to strengthen the overall structure twenty-four gussets were silver soldered in place around the chamber. These were nothing more than small triangular pieces of steel connecting the flange and the motor outside wall.

The silver soldering, at this point, became one of the most difficult parts of the manufacture. Warpage, leaks, and split joints all occurred and complicated the manufacturing problem. In all, the jacket was resoldered five times before the leakage was stopped.

At this same time, the injector was being produced. The original feed design called for the fuel leaving the jacket to be sent out through pipes to a fuel manifold on top of the injector. This was replaced by a system whereby the fuel bleeds directly from the jacket into the injector internal to the jacket wall.

Water flow tests of the engine revealed several leaks around the silver soldered joints. These joints were redone until leaks were no longer present. With the assembly of the injector, the engine was ready for testing.

While the engine was being constructed, static test facilities were also being built in the Rotor Lab of Rickover Hall. An "I" beam thrust stand was installed and instrumented with pressure transducers and a strain gauge thrust bar.

Power, chart recorders, indicators, plumbing and the engine were fitted to the stand.

A water cooled flame deflector was also built to keep the flame from destroying the floor of the lab and to eject the smoke from the building. The deflector was of welded steel construction and cooled by water injected through a network of pipes inside the flame deflector.

From early September, when construction on the test stand began, solenoid valves of sufficient size and pressure rating could not be found. There were no valves commercially available that would deliver the needed flows at 300 to 500 psi. Finally some were found and purchased. However, these were only N_2 valves for supplying pressure to the tankage, etc., and they were only rated at 300 psi. Liquid oxygen valves, relief valves, regulators and filters were still needed to complete the set up.

Since this type of specialized equipment is not easily obtained, a contact was made at a place where such equipment was likely to be found. A phone call was made to a Mr. Robert Odem at Rockwell International, Rocketdyne Division, in California. Through Mr. Odem, arrangements were made to have all of the equipment necessary for the static test stand shipped back to the Naval Academy. Mr. Odem and several members of his team provided tremendous support both in the areas of equipment and technical advice. A trip to California was arranged to discuss the project with Mr. Odem and others of

his staff, and at this time the engine design discussed in the preceding pages was analyzed. The design was found to be good, but, due to losses not originally counted on the thrust would probably be on the order of 940 pounds instead of 1000. Losses would occur due to the short chamber length, small throat radius and sharp corner at the chamber-converging nozzle junction.

After the meeting in California with the Rocketdyne people, the final plumbing diagram for the static test stand was generated. All the necessary valves, reliefs, filters, etc., had been shipped and were on their way to the Naval Academy. The shipment arrived on 6 November, and by mid-December all was in readiness for a static run on the engine. A diagram of the system can be seen in Figure 3.

While this was going on, feelers were sent out to Wallops Island, Virginia, and White Sands, New Mexico, as possible firing locations for the flight vehicle. Both places were encouraging about the possibility, so Wallops was set as the probable launch site due to its proximity to the Naval Academy.

The final preparations were made for the static firing in January 1976. The liquid oxygen tanks and lines had to be "super" cleaned for lox service in order to prevent internal explosion. They were vapor degreased and Freon cleaned at the Naval Research Laboratories in Washington. Liquid oxygen was procured from a local welding company, and the final calibrations of transducers and strain guages were done.

The regulators that supply pressure from the N_2 bottles to the propellant tanks were set according to the calculated pressure drops. Lox supply pressure was set at 300 (P_c) + 80 ($\Delta P_{injector}$) + 10 ($\Delta P_{plumbing}$) = 390 psia. Kerosene supply pressure was set as 300 (P_c) + 80 ($\Delta P_{injection}$) + 138 (ΔP_{jacket}) = 518 psia.

During the week preceding the test two friends of the author, Mr. Steve Brown and Mr. Nick Kirchner, arrived from California to help run the static test. The author and these two men had spent much time in high school designing, building, and firing rockets in the Mojave Desert in California. They were therefore both experienced in this type of work and were of great help in the testing that occurred. As well as these two individuals, Midshipmen Randall Parman and Alfred Manzi were instrumental in the success of the project. Their help in both the construction and testing phases of the project were invaluable.

The first static firing was set for 23 January 1976. Both high and low speed cameras had been installed in the test lab to record close up the engine burn. The engine was to be ignited by firing a pyrotechnic device, mounted in the chamber, just prior to admitting the propellants into the chamber. The kerosene had to travel through a longer plumbing system and the cooling jacket, but it was required that the fuel precede the oxidizer into the chamber to ensure a smooth start. For this reason the kerosene valve was to be opened

two seconds before the lox valve was opened.

The entire sequence was as follows. T-60 seconds, the low speed 8 mm cameras were started. T-45, the pyrotechnic igniter was fired. T-30, the lox and kerosene tank vent valves were closed. T-25, the propellant tank pressurization valves were opened and the tank pressures checked by watching the chart recorder readouts from the tank transducers. T-20, the tank valves on both the lox and kerosene tanks were opened, filling and pressurizing the propellant lines up to the engine valves. T-15, the valves which supplied cooling water to the flame deflector were opened. T-2, the kerosene engine valve was opened. 0, the lox engine valve was opened and the high speed camera was started.

The count went normally until kerosene engine valve opening at T-2 seconds. The pyrotechnic igniter had been ignited at T-45 and was burning well, but the kerosene had arrived in the chamber much faster than had been anticipated and it drowned the igniter. The lox then reached the chamber and the combined liquid flow pushed the igniter assembly out of the nozzle. No ignition occurred.

The propellants were mixed and blown unburned down into the flame bucket. Such an occurrence presents a tremendous hazard of explosion since a hydrocrabon fuel jelled with liquid oxygen is worth several times its own weight in the explosive force of TNT. The danger in this case was lessened by the water deluge inside the flame deflector washing away

the propellant mixture. The failure of this test was attributed to the method of ignition and bad sequence of propellant flow to the engine.

By working through the night on 23 January, the problems encountered were solved and a new test set for the following morning. The valve sequencing was corrected by slaving the lox valve opening to the open-closed indicator on the kerosene valve. This provided for less than a .1 second delay between the kerosene valve actuation and lox valve opening. The igniter was redesigned and a spark generator was used to power what was designated a "magic wand." This "magic wand" consisted of a long copper rod on one end of which was soldered a copper ring held in place by spokes radiating from the central rod. This rod was inserted through the nozzle and held the ring approximately 5/8 of an inch from the injector face. The entire igniter was insulated from the engine by a micarta cross bar clamped to the exit nozzle. By grounding the engine and hooking the "magic wand" to the spark generator, an annular ring of electric sparks was produced between the wand and the injector face. The diameter of the copper ring corresponded to the diameter of the inner row of injector holes. Around the central rod of the wand was wrapped a slow burning pyrotechnic which was also ignited before propellant valve opening.

On Saturday, 24 January 1976, the propellant tanks were refilled and a second static test attempted. The count went

smoothly and the firing was a complete success. The engine produced approximately 900 pounds of thrust for 13 seconds with a chamber pressure of 280 psia. These were excellent results for a first run.

Upon inspection, the engine proved to be in excellent shape. There was no erosion of any of the internal surfaces and the chrome plating on the injector (done to prevent rusting of the lox manifold) had not even discolored. Within one minute of engine cutoff the engine could be touched with a bare hand without discomfort, which attested to the success of the cooling system.

To ensure that the success was as complete as it was thought to be, a second hot firing was scheduled for the following Monday. The weekend was used to recalibrate and work some of the bugs out of the test stand instrumentation. Following the same firing procedure, a second completely successful run was obtained. Again, the thrust was approximately 900 pounds at a chamber pressure of 300 psi for a duration of 12.3 seconds. The burn time on both tests was under the original specification of 15 seconds due to lox boil off and lack of sufficient lox tank volume. This will be explained in the next section.

The exact thrust-time P_c - time curve for this static test can be seen in Figure 4. Again in this test, the engine sustained absolutely no erosion or deterioration. The engine could now be disassembled, cleaned, and readied for incorporation

into the flight vehicle. Having completed the testing, it was odd to reflect on the fact that the entire static test stand, the product of five months of work, was built to test fire the engine for a total of 26 seconds.

Analysis of the engine firing data gives the following results. Specific impulse is equal to

$$I_{sp} = \frac{c}{g} = \frac{F}{\dot{W}} = 212 \text{ sec}$$

Therefore, the effective exhaust velocity, c, is

$$c = I_{sp}\ g = 6814 \text{ ft/sec}$$

The total impulse I_t equals

$$I_t = F \times t = 11{,}700 \text{ lb/sec}$$

The characteristic exhaust velocity, c*, equals

$$= \frac{I_{sp}\ g}{C_F} = 4845 \text{ ft/sec}$$

At any rate, the test stand and all its related equipment were dismantled and work was started on the flight vehicle. Tanks of the required volume were designed and constructed. Due to the lack of the metal spinning equipment needed to make hemispheresfor the tanks ends, the tanks were built with flat end plates. However, during testing these tanks ruptured at less than one-half their designed pressure. All failures were along welds.

Since it looked unlikely that tanks could be quickly produced, the decision was made to try and acquire commercially available tanks. Stainless steel breathing oxygen tanks of

approximately the correct volume were obtained. However, these tanks were far from ideal. They were 2 inches too small in diameter and the lox tank was not as large as it should have been for a 15 second burn. However, there were no alternatives except to design around the tanks and use them.

The high pressure supply tanks were the next to be designed. Using a simplified analysis, the required weight of N_2 is defined by the equation

$$W_g = \frac{p_p V_p}{RT_o} \frac{k}{1 - \frac{p_g}{p_o}}$$

where p_p is the average propellant tank pressure, V_p is the propellant tank volume, R is the gas constant k is the specific heat, p_g is the tank pressure plus the minimum Δp across the regulator, and p_o is the initial supply tank pressure. For this case W_g = 4.8 pounds N_2 at 3000 psig.

The required volume, V_g, is

$$= \frac{W_g \; RT_o}{p_o} = 568 \text{ in}^3$$

Lacking the necessary welding and forming equipment, no attempt was made to manufacture the high pressure tank. The only commercially available tank found with the required volume weighed 33 pounds. A smaller gas bottle with a volume of 270 in^3 and a weight of 10 pounds was found, but the volume was too small. This bottle could be used as the pressure bottle if the rocket was allowed to go into a blow down mode. In such a system, the gas bottle provides regulated pressure

until the supply pressure is equal to the regulated pressure. From that point there is a chamber pressure and thrust decay since the only force expelling the propellant from the tanks is the expansion of the gas from regulated pressure down to the ambient.

To resolve the problem of whether to use the heavy or the light gas bottle, an analysis of flight performance was done balancing the problems of thrust decay versus weight. This analysis showed the blow down mode as giving the higher performance.

However, before much hardware had been generated for the blow down mode system, some spherical hydraulic accumulator tanks were found. These tanks were 7 inches in diameter and weighed 4.5 pounds each. Four such spheres had a combined volume of 588 in^3 and a weight of only 18 pounds. The use of these tanks would cause some plumbing complications, but would deliver full pressure to the tanks for the duration of the burn, and the weight was low. The decision was made to use them and the airframe design was begun.

The two stainless propellant tanks, four pressure spheres and the engine were incorporated into an airframe consisting of 4 longitudinal stringers and 10 circular ribs. The stringers were manufactured of .5" square aluminum tubing. The ribs were made of 8" diameter aluminum disks with the center cut out to accommodate the tanks. A stiffening flange 3/4" wide was welded around the edge of each rib and gussets were riveted to it to tie into the stringers. The lower end

of the stringers terminated in the engine mount ring. The engine was held to this ring by 12 1/4-28 bolts. Above the pressure bottles a 10" section of the airframe was left empty for the payload cannister and camera pod. Originally this was to be the end of the rocket and the nose was to be attached here. However, it was discovered that a 24 foot diameter parachute would not fit in just the nose cone, so an additional section one foot long was added above the payload to allow room for the parachute.

The nose itself was specially designed for parachute deployment. Two fibreglass halves were manufactured and stiffened internally with an aluminum framework. The nose tip was made of aluminum and was also split in half. In the center of the tip, a cavity was machined to hold an explosive charge. In flight, the nose would be held together by a plastic bolt. When the nose charge was detonated, the bolt shears, the nose halves split and fold back, allowing the parachute to deploy. This nose was hinged to the forward end of the parachute tube.

The aluminum framework of the missile was to be covered by a 0.016" thick 7075-T6 aluminum skin riveted in place. Four access doors were left through the skin for propellant loading, pressurization and payload adjustment. Fins were attached to the engine and the stringers with screws. The airframe was configured with a boat tail down near the engine by fairing in the skin from the 8" missile diameter to the

5.5" diameter engine exit ring.

However, before the skin could be applied to the rocket, all plumbing, wiring, testing and internal construction had to be completed. Tanks and plumbing were laid out as follows. Near the forward end of the rocket the 4 high pressure nitrogen bottles were manifolded together and plumbed into a manual valve. The outlet of this valve leads into a 3000 psi solenoid valve. Downstream of this valve the pressure line is split and runs to both the lox and kerosene tank regulators. The low pressure lines downstream of the regulators run to their respective tanks and discharge into gas diffusers to prevent nitrogen from being shot down into the propellants. There is also a check valve in the lox pressure line to prevent oxygen from backing up into the regulator and then possibly into the kerosene tank. There is also a lox vent valve on the lox tank which will be discussed a little later. Special fittings were manufactured for the lower end of both the lox and kerosene tanks to ease the plumbing problems and to provide a point to which the loading valve stubs could be welded. The propellant lines extended from these fittings down to the engine. Burst diaphragms were installed just downstream of the tank fittings. When the high pressure solenoid valve is opened, the burst diaphragms are ruptured and the propellants flow to the engine.

The lox vent valve posed a major problem. No commercially produced valve could be acquired for less than $800. Therefore, A pneumatically operated valve was designed and built. The

valve was activated with high pressure nitrogen from the N_2 bottle. During testing at normal temperature the valve worked perfectly, but at liquid oxygen temperature there was a possibility of the "O" ring seals breaking down and the valve leaking. For this reason, the valve was scrapped and a new one built by modifying an existing fuel valve. This valve was disassembled, new parts were designed and manufactured, teflon seals were made and an explosive actuator was added. This valve proved to be less than one tenth the weight of the originally designed valve and it worked flawlessly.

With the completion of the internal valve, indicator, and umbilical wiring, and installation of the lox vent valve, system flow tests could be run. After exhaustive testing of valves, burst diaphragms, parachute cables, explosive actuators, line cutting igniters, etc., the rocket was run through a complete flow and system test. The few problems that were uncovered were remedied and the entire rocket disassembled to reclean the tanks. After cleaning and reassembling the skin and fins were mounted completing the airframe. A center of gravity center of pressure stability analysis was performed on the complete rocket, which placed the CP 65 inches forward of the nozzle exit, which is 41 inches behind the empty CG and 33 inches behind the loaded CG. This situation promised good stability in flight. The complete vehicle is 18 feet 3 inches long, 8 inches in diameter, weighs 106 pounds empty and 204 pounds loaded with fuel. The empty CG is 106" from the nozzle exit and the loaded CG is 98" from the exit.

The recovery system for the rocket was designed to bring the entire vehicle back intact. The parachute ejection system is simple. In the nose tip is mounted a light dependent resistor (LDR). This device can differentiate between sky and ground on the basis of light intensity. The LDR is coupled with a relay so that uponreaching peak the rocket turns over, the LDR sees the ground and trips the relay. The relay fires the nose charge and deploys the drogue chute. Since the rocket has a projected peak altitude of over 50,000 feet, deployment of the main at peak is impractical. Therefore a drogue, designed to bring the rocket down at 80 ft/sec, is deployed at peak and is tied to both the rocket and the top of the main chute. The tie line to the rocket runs through an explosive line cutter and until this cutter is fired the drogue cannot deploy the main chute. As the rocket passes through 10,000 feet on the way down, a barometric switch fires the line cutter and deploys the main. The barometric switch cannot fire the line cutter on the way up, because of a cutout switch which requires tension on the drogue tie line in order to close.

The payload built by the author consisted of an on board camera system consisting of two 8 mm cameras, looking down the length of the rocket through externally mounted prisms. One camera is started just prior to launch via the control panel and umbilical cord and films the flight on the way up. The second camera is actuated by tension on the drogue tie

line so it will begin taking film at peak and continue filming on the way down.

The payload also included a gamma radiation experiment built by Midshipman Theodore Fredrick. Information gathered from the gamma detector was telemetered back to earth via an s-bank transmitter operating at 2259.5 MH_z. The rest of the payload space is taken up by the LDR system, barometric switches and batteries.

During the time the rocket was being constructed, plans for a launch at Wallops Island fell through. Therefore, arrangements were made through Commander Sharp and Mr. Lloyd Briggs to fire at the Naval Ordnance Missile Test Facility, White Sands, New Mexico. A firing date of 22 April was tentatively arranged, but was later cancelled and reset for 11 May.

The missile was finished and crated for shipment on 4 May 1976. The author and Mr. Art Goehring, a machinist from the Technical Support Department in Rickover Hall, traveled to White Sands on 5 May. Mr. Goehring had been connected with this project since September and was invaluable to its success both while the rocket was being built and during the launch.

The first few days at White Sands were spent gathering up the necessary equipment and interfacing the control panel built for the rocket with the existing launch and umbilical systems. The rocket arrived on 7 May and reassembly was begun the next morning. By Sunday, 9 May, the rocket was ready for check out on the pad. Liquid oxygen was delivered

and 10 gallons of JP-4 were procured and stored in the baricade structure out at Launch Complex-36. An umbilical wiring problem developed and took most of a day to clear up. The cables between the launcher and the blockhouse were shorted by some new construction. Once the problem was discovered, the umbilical was shifted from the right to the left arm of the launcher. The missile was run through a simulated launch and checked out fine.

Monday, 10 May, the rocket was fitted with launch lugs and placed on the rail. By Monday, Randy Parman, Ted Fredrick, Steve Brown, Art Goehring, and Emanuel Crisalli were all busy working to get the missile and ground support equipment ready for the 0930 launch on Tuesday, 11 May. GMM1 Whitby and BM1 Busby from the Research Rockets Branch at White Sands were also working on the launch.

This crew arrived out at Launch Complex (LC)-36 by 0600 on the morning of 11 May. The rocket was rolled out of the barricade structure and placed on the rail. An umbilical check and quick LDR check were run. The nose and payload section were assembled and placed on the rocket.

At 0730, pressurization of the N_2 spheres was begun from a 6000 psi source. Immediately a problem developed as nitrogen leaked past the closed manual valve and was coming out of the lox and JP-4 vents. The leak was slow, but would have ruptured the burst diaphragms had the tank vents been closed. The valve could not be removed to be repaired, so the N_2

solenoid was energized and closed, which stopped the leak to the tanks, and pressurization was continued. The manual valve was left closed to prevent a major pressure drop through the tanks in the event of a solenoid failure.

The N_2 bottles were filled to 3100 psig and the launcher arm was elevated to 82^o. Kerosene (JP-4) fill was started immediately. Parman, Busby, and Whitby fueled the JP-4 while the lox feed lines were being rigged.

While the fueling was going on a problem developed in the recovery system. The LDR was shorting to ground and cutting out the recovery system. The problem was remedied by bypassing one of the circuit connections. The lox tank was being filled at this time.

At T-4 minutes the N_2 fill and lox fill lines were broken down and the access doors on the rocket buttoned up. The last person left the pad at T-60 seconds and counting.

At T-30 seconds the button on the control panel for lox vent closure was pushed. Immediately the lox vent indicator light came on green indicating lox vent closure. At T-5 sec the on board camera was started and the spark igniter was turned on. At T-3 the loud speaker voice stopped counting and began to repeat delay, delay. There was no explanation for the cessation of the count, but after approximately 20 seconds the fire button was pressed and nothing happened. There was no explanation for the failure, but the missile had to be disarmed and the lox vent valve reopened.

The vent was opened and the missile examined to find the

cause of the problem. Kerosene was dripping from the nozzle, which seemed odd, since if there had been enough pressure to rupture the diaphragm, why was the kerosene tank still full? Eventually it was discovered that the manual valve in the nitrogen system was not open. A faulty valve stem had prevented the valve from opening. The kerosene diaphragm had been ruptured only slightly by the leakage of pressure past the manual valve when the fire button had been pushed to open the N_2 solenoid.

Once the problem was discovered, it was relatively simple to repair. The kerosene tank had to be drained, the diaphragm replaced, the tank refilled, the vent valve charge replaced, and the N_2 spheres topped off to bring them back up to 3100 psig. Once this was done and the manual valve opened, the rocket should be ready to go.

A second firing time was scheduled for 1500 on the same day. The rocket was ready to go by 1300. At 1420 the lox tank was topped off and by 1450 the rocket was buttoned up and the crew was on its way back to the blockhouse. At T-30 the lox vent was closed and the indicator came on green. T-5 the camera and spark generator were started. At zero the fire button was pressed and again there was no ignition. The kerosene was dumped this time, but no liquid oxygen was pumped to the chamber. The second failure in as many times.

The rocket had to be disarmed again and the propellants drained. There was no explanation for the failure and no possibility of trying to launch again that day. There was

no possibility of sliding the launch to the next day either, since final examinations back at the Academy demanded the author's presence. The project looked doomed to failure.

However, the following morning it was discovered that another launch window was opening up on the 17th and 18th of May. After some very hasty phone calls on the morning of 12 May, the following plan was put into effect. While the author returned to the Naval Academy to muddle through final exams, Art Goehring would remain at White Sands to reassemble the rocket (which had been partially dismantled) and to set up for a third launch attempt the following week. It was discovered that the reason behind the failure of the liquid oxygen to reach the combustion chamber on the last launch attempt was that the lox burst diaphragm had been inadvertently doubled. Thus it did not burst when the nitrogen valve was opened.

This situation was corrected and the rocket was prepared for another test. On 15 May the author, having stumbled through finals, returned to White Sands. The rocket was ready to go, but the payload and launch system needed to be reworked. Over the weekend the original launch crew was reassembled minus Parman and Fredrick and plus Dr. A. Pouring. By midnight Sunday, 16 May, the missile had been checked out on the umbilical, the lox and kerosene diaphragms had been replaced, a new igniter had been made and checked out, and a new charge was installed in the lox vent valve. Launch time was set for 1115 the following morning.

The crew arrived back out at LC-36 at 0630 on the morning of 17 May. An LDR check was made, the nose and payload were assembled and the rocket was put back on the rail. A check was run on the spark generator and some problems were discovered but were quickly solved. There was not that much to do on the rocket and there was plenty of time, so there was no rush getting the rocket fueled. About 0900 the kerosene fueling began. Immediately problems developed; as kerosene was added to the tank the engine began leaking. The kerosene diaphragms were bad and had to be replaced. The tank was drained, new diaphragms installed and the tank reloaded. However, before there were five pounds of kerosene in the tank the fuel came dribbling out of the engine again, so the procedure of draining, replacing diaphragms, and refilling had to be done all over again. To change the kerosene diaphragms was a very lengthy and difficult procedure. This, coupled with the fact that it had to be done four times before the leaks were stopped, put the fueling operation way behind schedule. At T-30 minutes the N_2 bottles were not yet under full pressure, the lox loading had not even begun and the kerosene leaks had just been stopped. At T-15 minutes lox loading began. At T-5 minutes a hold in the count was initiated. A total delay of 15 minutes was needed to pressurize, finish loading lox and break down the fill plumbing. At T-3 minutes, everyone was on the way to the blockhouse.

Everything looked go. The count continued normally down

to T-20 seconds when lox vent closure was initiated. The indicator that shows the valve was closed did not come on. The count was held again at T-18 sec. The valve had to be closed manually, and the easiest way to do so was through the lox vent valve charge access panel. Back out on the pad, the access panel was pulled off, the charge removed and a drill punched through the charge hole to trip the vent valve.

It was difficult to tell if the valve had closed down, but a voice came over the loud speaker saying "we have a green board." The pad was cleared again and the count resumed at T-3 minutes.

The excitement was even more intense now as people waited to see what would happen. After two complete ignition failures and the vent valve problem that morning, nerves were at maximum tension. The count droned on--at T-5 the camera was started, T-3 generator on, 2, 1, FIRE. Then through the foggy blockhouse window a glow could be seen lighting up the ground and launcher near the missile. The glow burst into a brilliant, powerful jet emerging from the tail of the rocket. The roar of the engine shook the ground all the way back to the blockhouse and the flame had steadied into a blinding white hot star beneath the rocket. It rose, slowly at first, with grace and ease. The burn and acceleration were smooth, and the rocket was truly a sight to behold as it left the rail--white against a deep blue sky with a long brilliant tail of flame.

The rocket accelerated rapidly, the flight was perfectly stable, and the rocket did not spin as it flew. The burn continued for its allotted length of 12.8 seconds. At this point, the rocket was passing through Mach 1 (verified by radar track) and was at an altitude of approximately 8000 feet.

In the next instant the nose charge detonated, blowing the nose halves off the rocket. The drogue deployed, snapped its tie down to the rocket and deployed the main. Deployment of a 24 foot chute at Mach 1 is healthy for neither the chute nor the rocket. The rocket was jerked broadside to the wind stream as the canopy opened and the canopy was immediately torn to shreds. As the rocket traveled up to about 10,000 feet the fins began breaking up and pieces were falling off the airframe. There was still some propellant left in the tanks during this maneuver, and after the rocket had lost the nose section, fins and all stability, the engine sporadically cut on and off. With the liquid oxygen exhausted and only a kerosene flame sputtering from the nozzle, the rocket fell horizontally back to earth and crashed about 3/4 of a mile from the launch pad.

Upon examination of the wreckage, the first thing noticed was that the lettering that had originally said "Beat Army" down the side of the missile was missing the forward section and now said "Eat Army." Further perusal revealed that although the on board cameras had been destroyed the film cartridges

were still pretty much intact. The engine was in perfect condition except for a bent fuel feed line. The rest of the rocket was severely altered in design. Let it suffice to say that although it was shipped to White Sands in an 18 foot crate, it could have been shipped home in a shoe box.

Analysis of the film of the launch may reveal more as to how the missile broke up, but an explanation as to why the nose charge detonated prematurely is lacking. The only plausible explanation is that during transonic buffeting the charge circuit wires could have been pinched and shorted setting off the charge. The only other explanation is that the LDR system closed the firing relay prematurely and the mercury switch (used as a safe and arm device ensuring a broken firing circuit until after burn out) was closed by the gyration induced by buffeting. However it happened, the fact that the charge had gone off was verified by recovery of one of the nose halves which had a sheared plastic bolt and burned powder from the charge. As of 1138 Mountain Daylight Time, 17 May 1976, the project had been completed as per the original proposal. A liquid fueled rocket had been designed, constructed and flight tested.

EPILOGUE

Well, the project was completed. The enthusiasm and excitement of the NASA and Navy people who had gathered there to see the rocket fly are beyond belief. From the cheering, shouts of joy, and hand shaking that went on after the flight, one would think Apollo 11 had just set down on the moon. The support, determination and indomitable spirit of the people at White Sands was a marvel to behold; and I am sure that if the engine had not fired, the rocket would have left the rail powered by the sheer will of the spectators.

One could look at the launch as a feeble success, knowing the fate of the rocket as it approached 8,000 feet, but all the months of work, frustration, anxiety and previous failures were outweighed by that 12.8 seconds of perfect flight. If I may digress from the realm of scientific reporting, only one who has seen a missile fly can understand the excitement and thrill of a launch. To see a polished metal machine, which for so long lay dormant in the laboratory, explode into vibrant, pulsating life and rise at fantastic speeds on a tail of blinding fire is an unforgettable experience. The roar of the engine tears at a man with a physical violence and to watch such a machine fight and struggle so hard to free itself from the bonds of earth is awesome.

These rockets are the simplest of machines in principal yet the most complex in application. Of all the different types of machines, they are the most unforgiving of design or construction errors. And, above all else, they fire the imagination of men because they are the first generation of

machines to take man beyond the realm of his own tiny planet.

As I stated in the beginning of this paper, this project has contributed nothing to the field of rocketry. There is little that one man and $800 can do to compete with NASA. But the lessons taught the individual by such a project could never be duplicated in any classroom. The energy to learn, design, build, test, fail and try again is generated by the anticipation of that roar, fire and 12 seconds of flight. The enthusiasm and drive necessary to work day in and day out, round the clock, to find solutions to problems and answers to questions, is found in the thrill of being able to turn ideas into working machines.

There comes a time for the black boxes and theory of the textbooks to be put into practice, and watching a machine obey the laws and follow the theory which went into its building generates an enthusiasm which demands more knowledge, more learning, and better machines.

The opportunity afforded me by this project has been the most fantastic academic experience of my scholastic career. Nine months of work for 12 seconds of flight does not sound like a very good trade, but it was more than I could ask for. Rockets, like this one, are but frail against the wonders of the stars, but they are the key that will unlock the door to the mystery of the heavens.

MISSILE DATA SHEET

Engine Data (Experimental)

Fuel	Kerosene
Oxidizer	Liquid oxygen
Thrust	900 lbs
I_{sp}	212 sec
Burn time	13 sec
Chamber pressure	300 psig
Cooling	Film and regenerative
Effective exhaust velocity	6814 ft/sec
Total impulse	11,700 lb/sec
Characteristic exhaust velocity	4845 ft/sec

Vehicle Data

Feed system	3100 psi GN_2
Weight N_2	4.8 lbs
Volume N_2	588 in.^{3}
Lox tank pressure	390 psi
Kerosene tank pressure	518 psi
Length	Approx. 219"
Diameter	8"
Weight of kerosene	30 pounds
Weight of lox	42 pounds
Empty weight	138 pounds
C.G. (measured from tail)	Approx. 98" (loaded)
	Approx. 106" (empty)
C.P. (measured from tail)	65"

Stabilization	Fins (4)
Fin area (each)	200 in^2
Recovery system	Parachute - drogue deployed at peak and main released at 10,000 ft.
Diameter of drogue	4 ft.
Diameter of main	24 ft.
Descent rate of drogue	Approx. 80 ft/sec
Descent rate on main	Approx. 10 ft/sec
Req. voltages for ground support gear	24-28 VDC, 115 VAC

Lox loading - Lox system is designed to be loaded from a commercial pressurized lox dewar-lox fill port is 1/2" "AN" flare fitting.

Launch rack - No launch lugs, etc., have been built into the missile - it was originally designed to be launched from a four rail tower.

Intended launch angle - 85 degrees

Telemetry - S Band - Transmitter is being acquired from NASA, Green Belt, Md. - Specific frequency unknown at this time. 2259.5 MH_z expected.

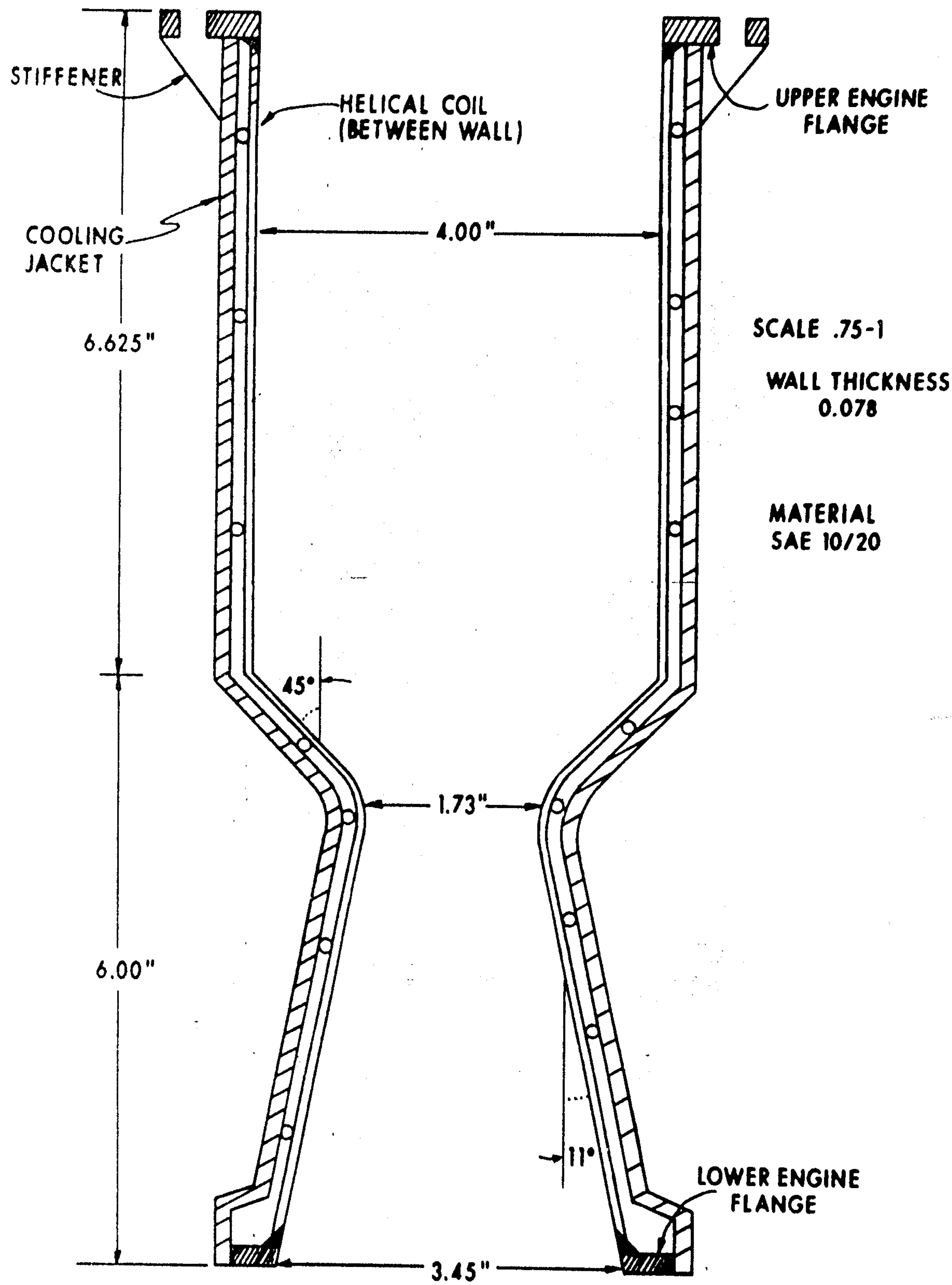

Figure 1

Engine and Cooling Jacket

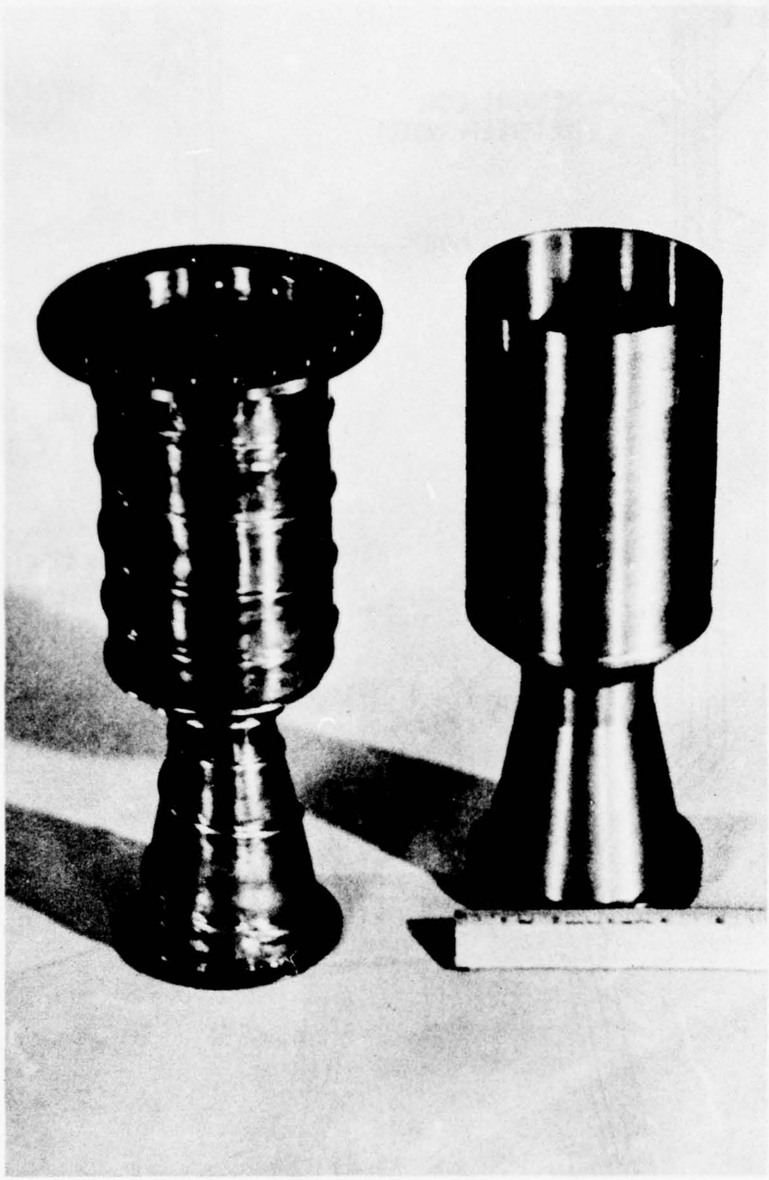

Figure 1a

Engine shell with cooling coil soldered in place and cooling jacket before assembly and welding.

Figure 1b

Engine installed in rocket showing fuel feed lines, welded cooling jacket, fin mount tabs, and injector flange stiffening gussets.

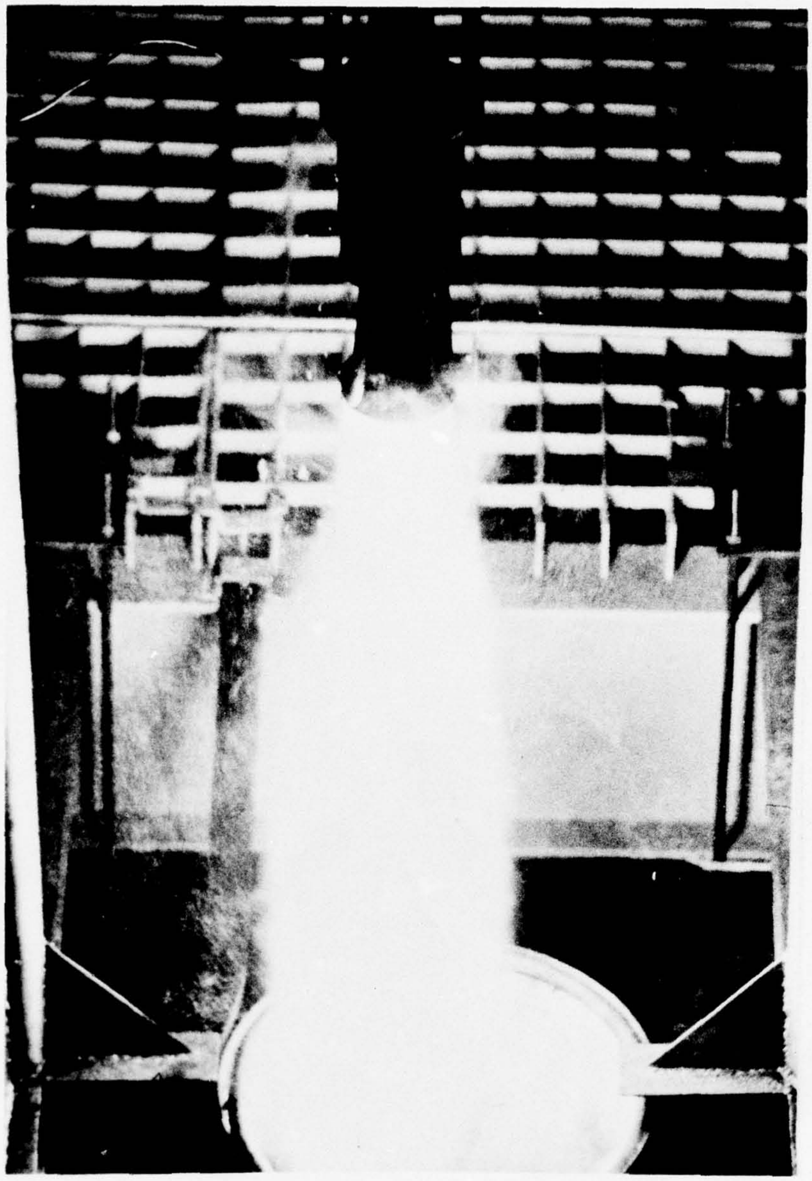

Figure 1c

Engine undergoing static test firing, January 1976.

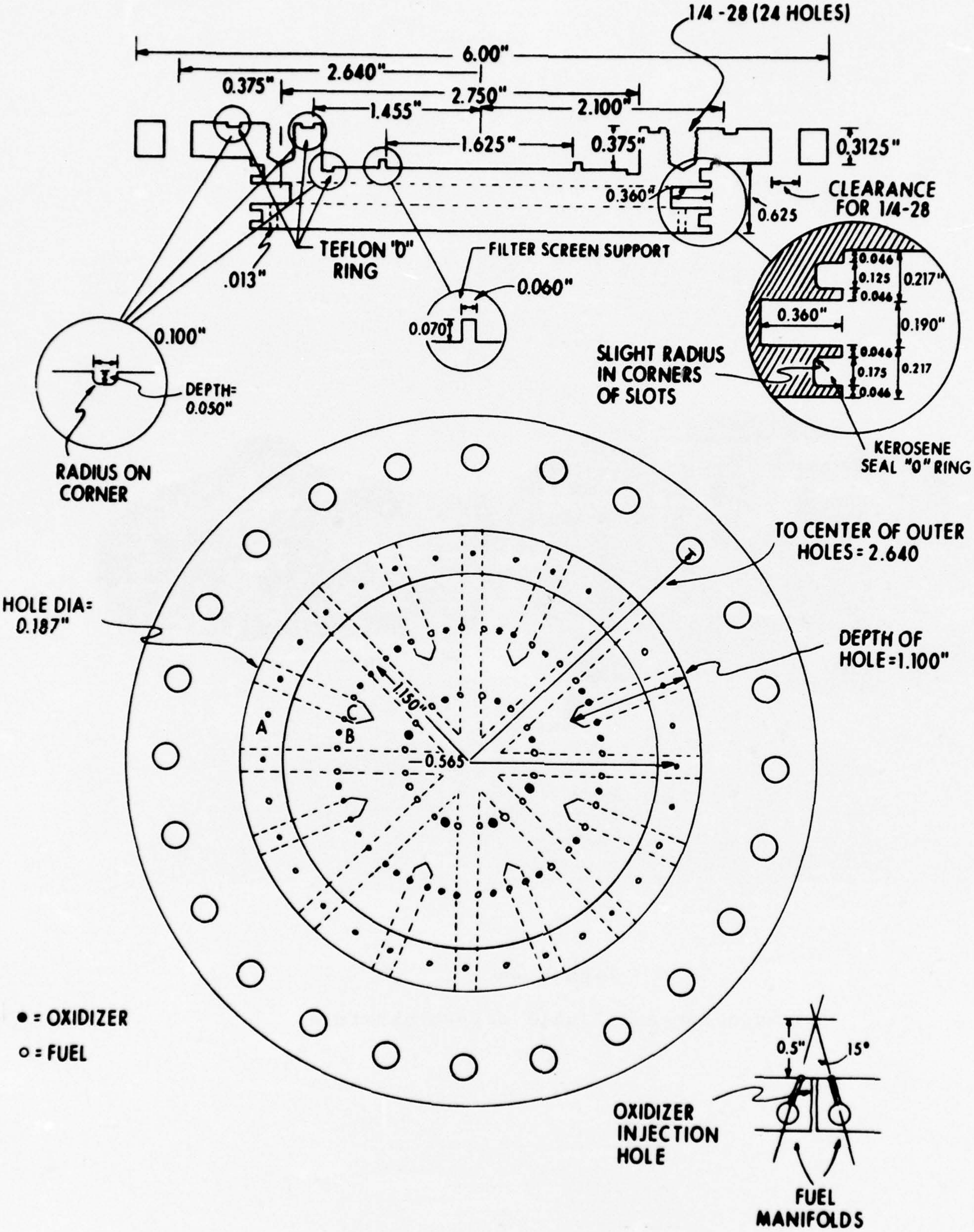

Figure 2

Injector

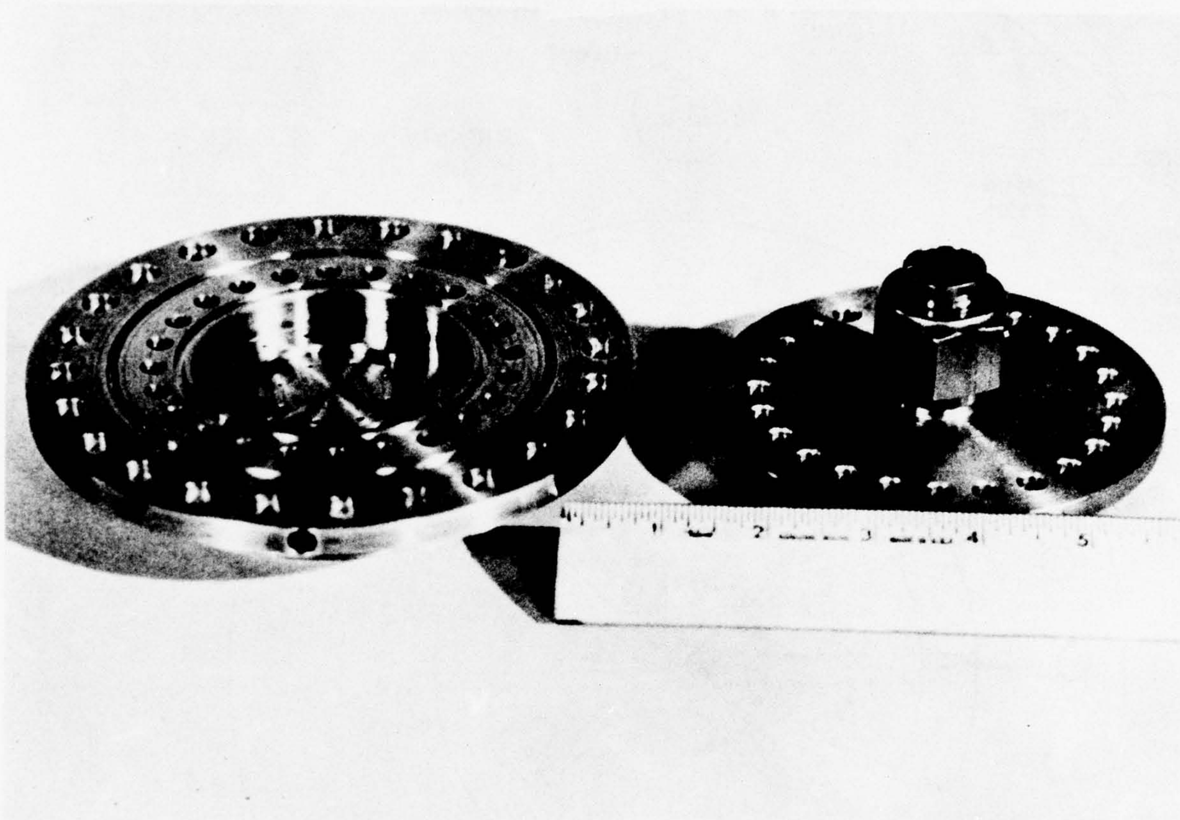

Figure 2a

Injector and liquid oxygen manifold

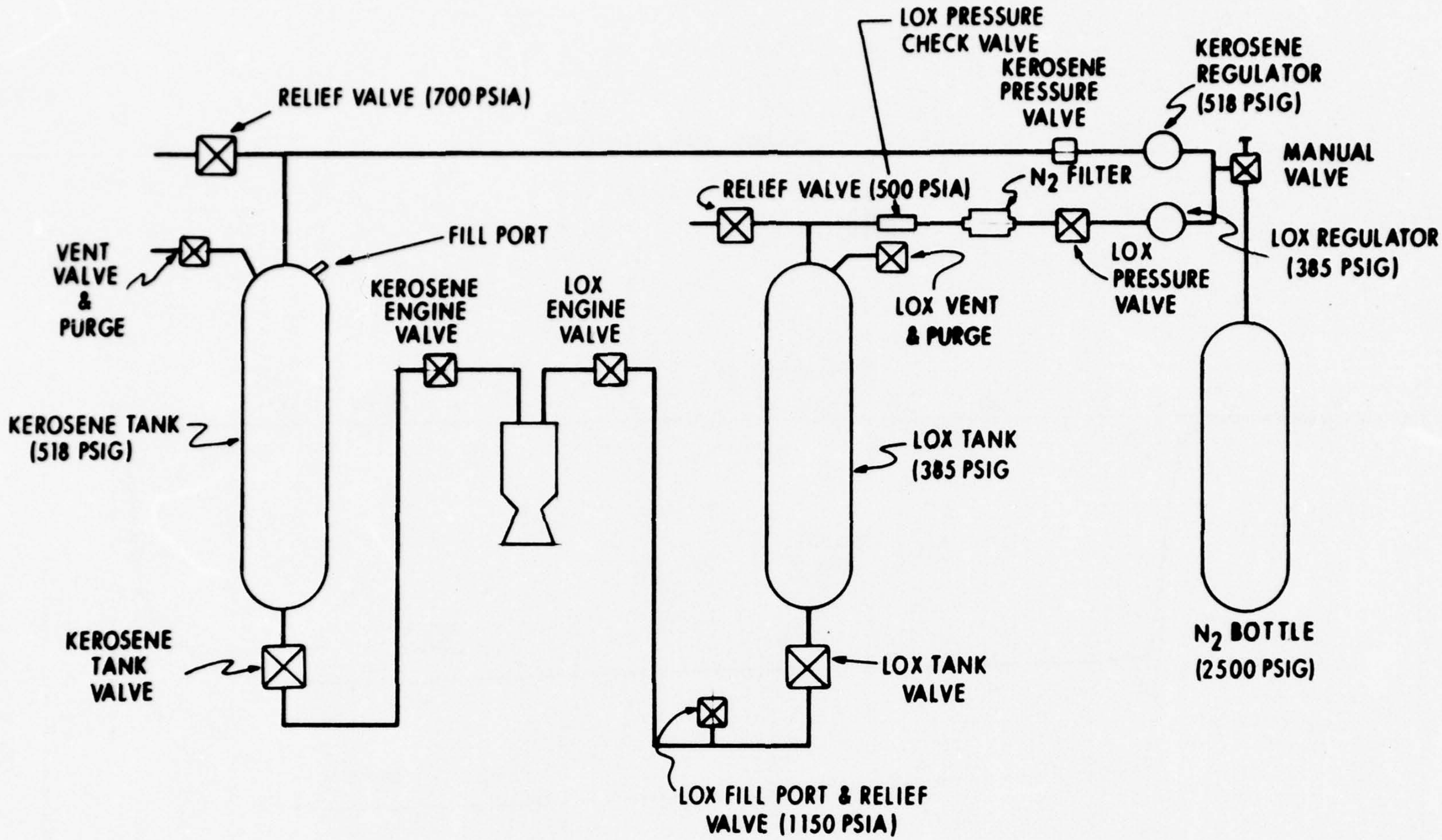

Figure 3

Static Test Stand Plumbing Diagram

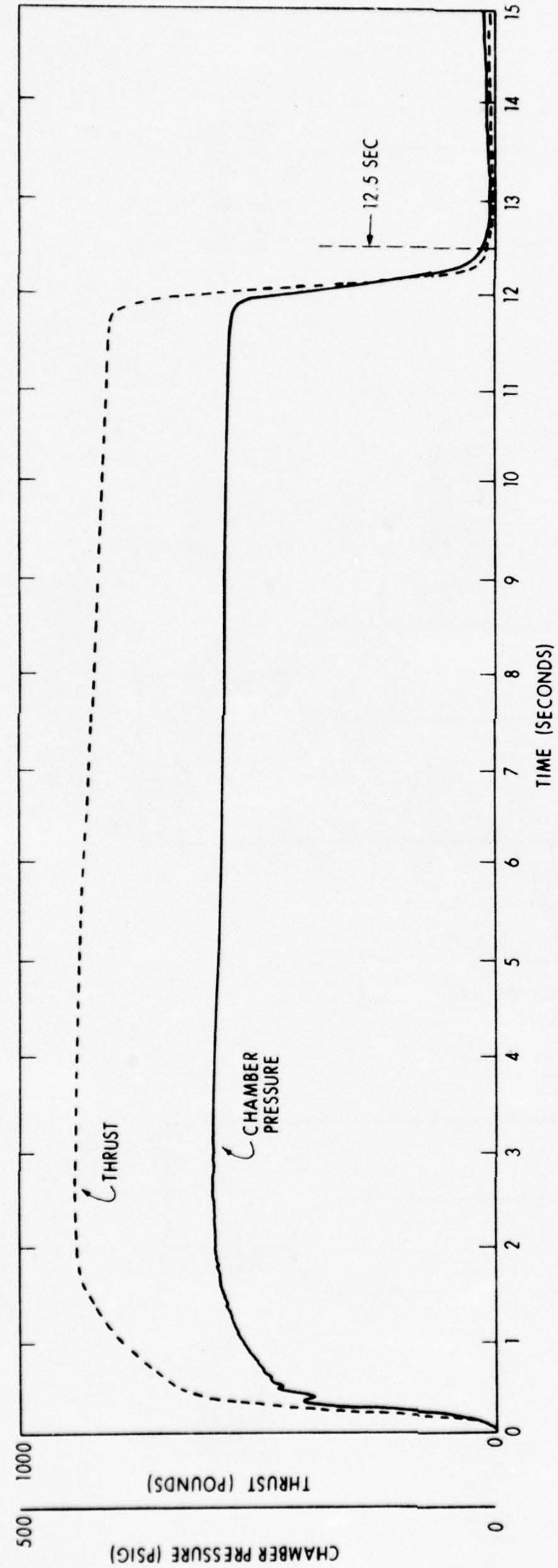

Figure 4

Thrust-Time Curve

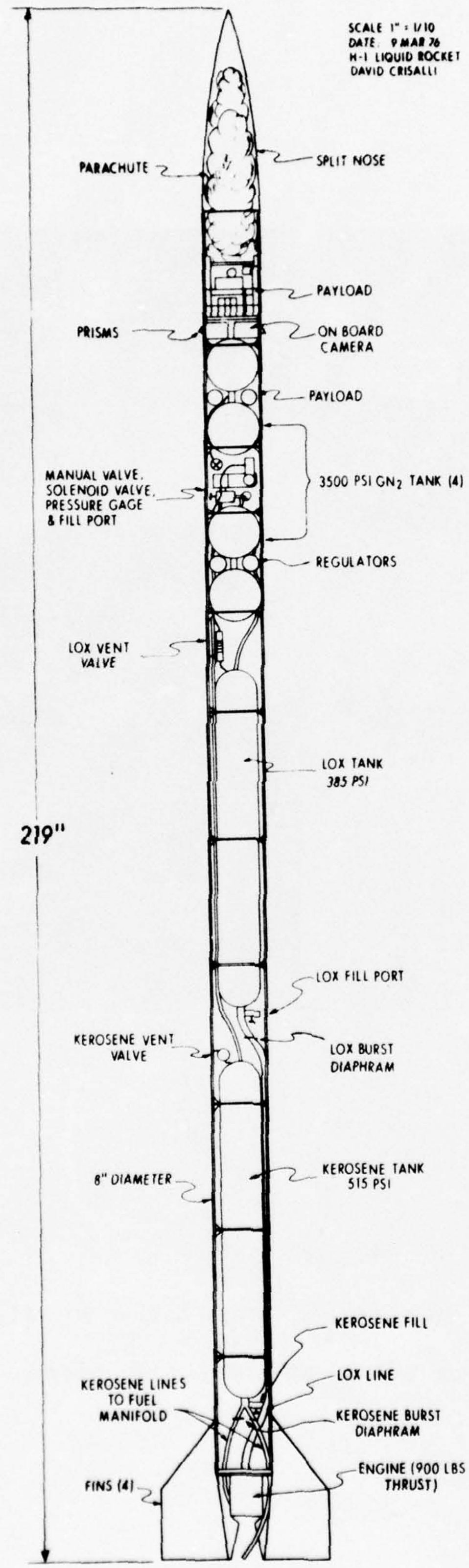

Figure 5

Liquid Rocket Details

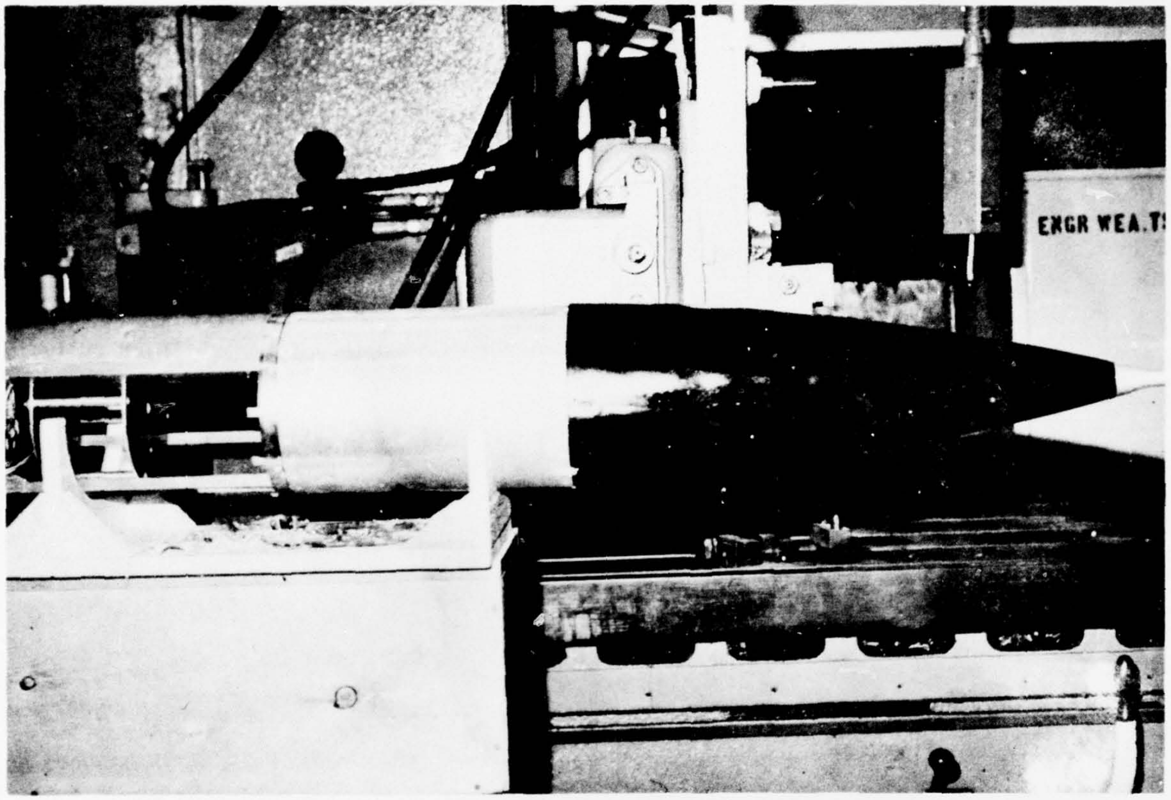

Figure 5a

Forward section of the rocket showing explosive nose tip, split nose cone, parachute tube, and payload section.

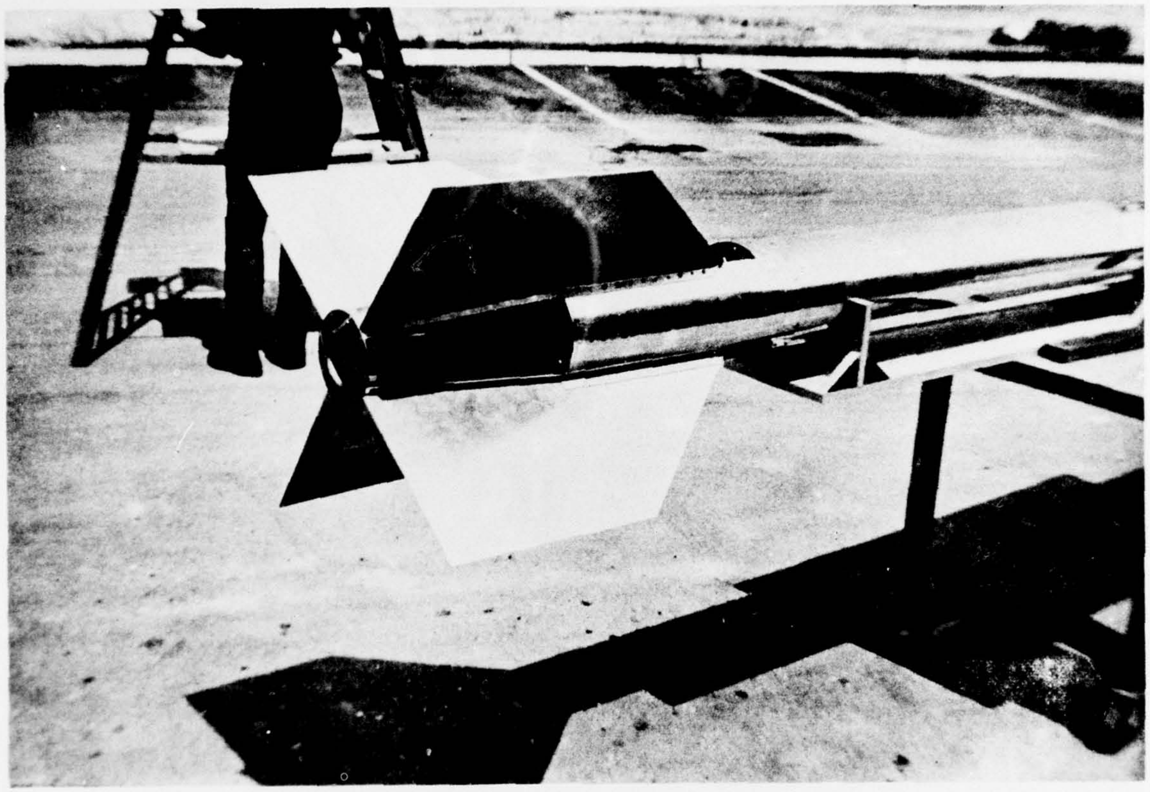

Figure 5b

Fin and Engine Assembly

Figure 5c

Rocket after final assembly on launch pad, LC-36

White Sands, New Mexico

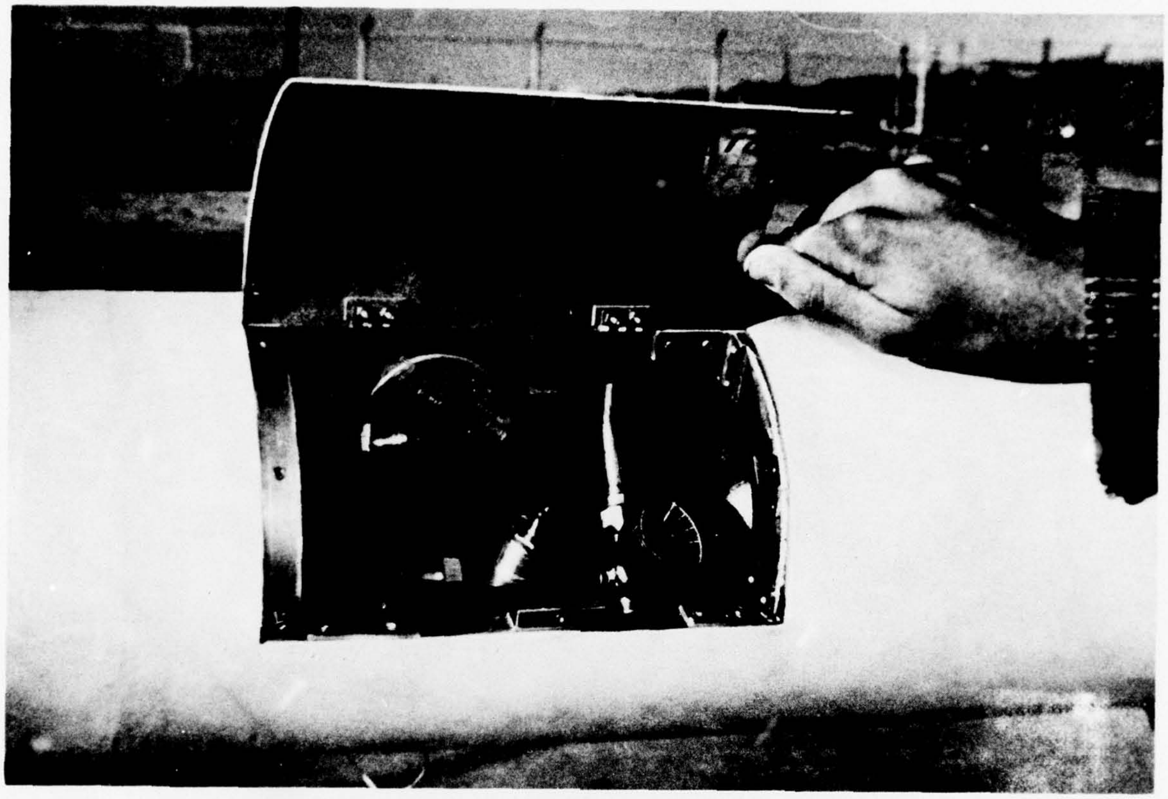

Figure 5d

High pressure plumbing and regulators

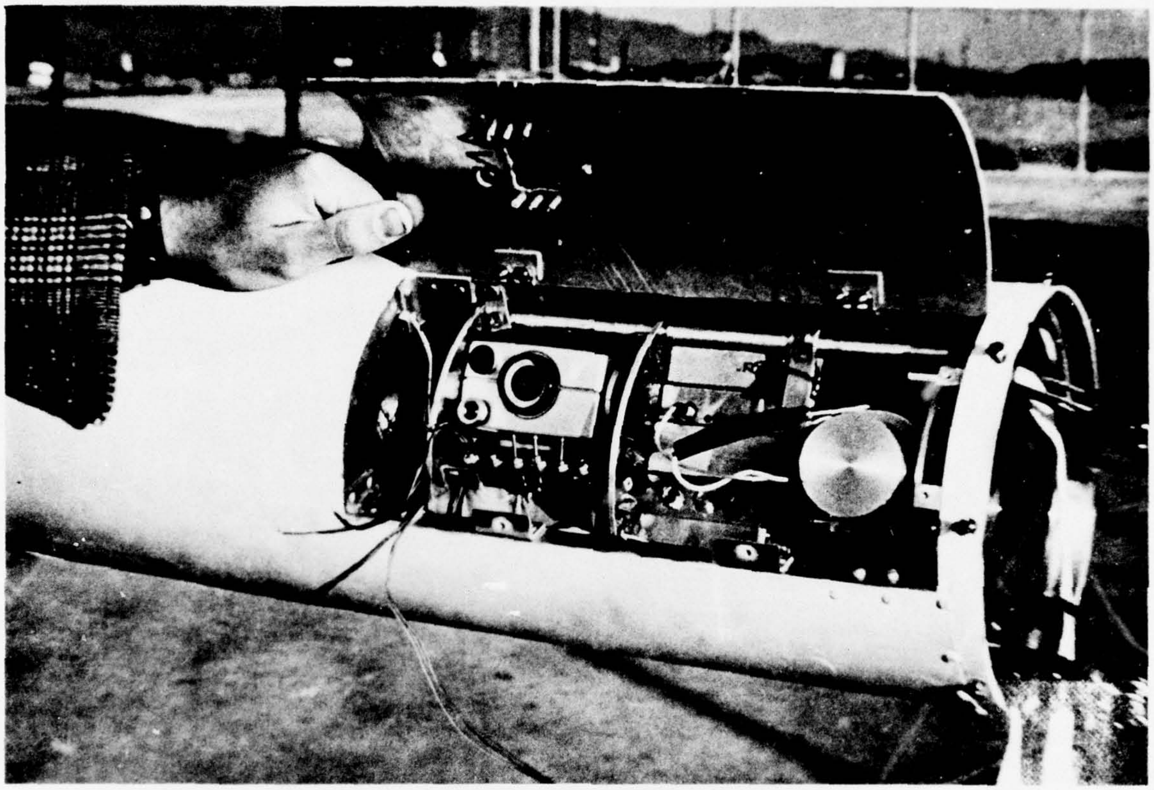

Figure 5e

Payload section. Prism mounted on door directs the view of the camera down the side of the rocket.

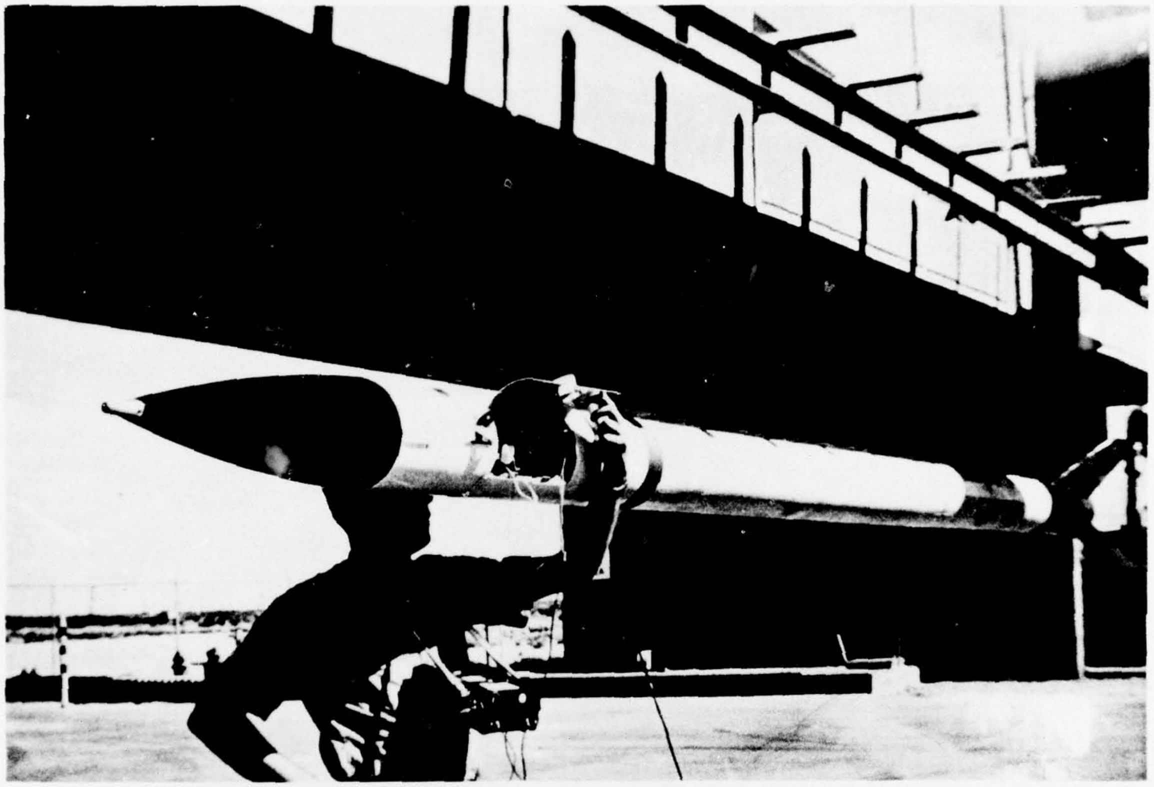

Figure 5f

Pre-launch umbilical checkout on the launcher. Pyrotechnic firing circuits are being monitored here.

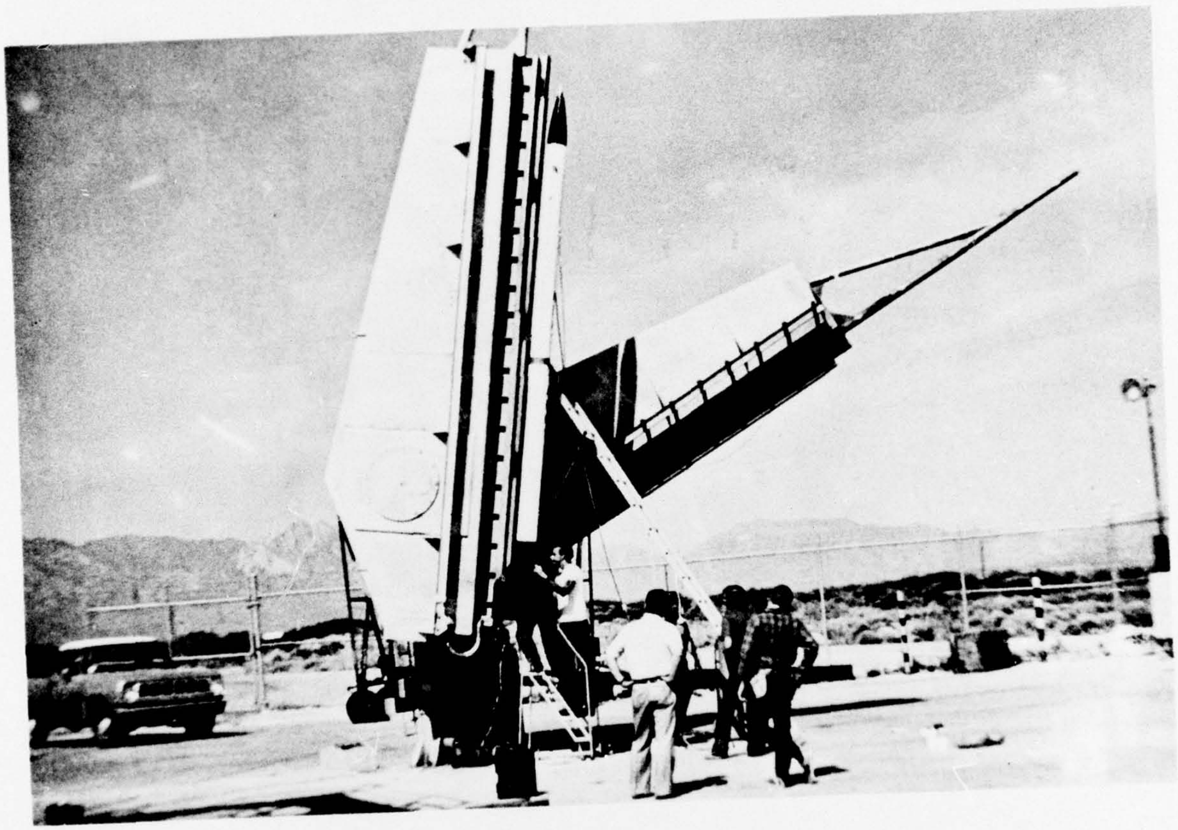

Figure 5g

Rocket on the rail and elevated. Leaking kerosene, burst diaphragms are being replaced for the third time.

Figure 5h

Kerosene fueling complete. Liquid oxygen fill is in progress and telemetry transmitter is being turned on.

Figure 51

The rocket ignites and leaves the rail.

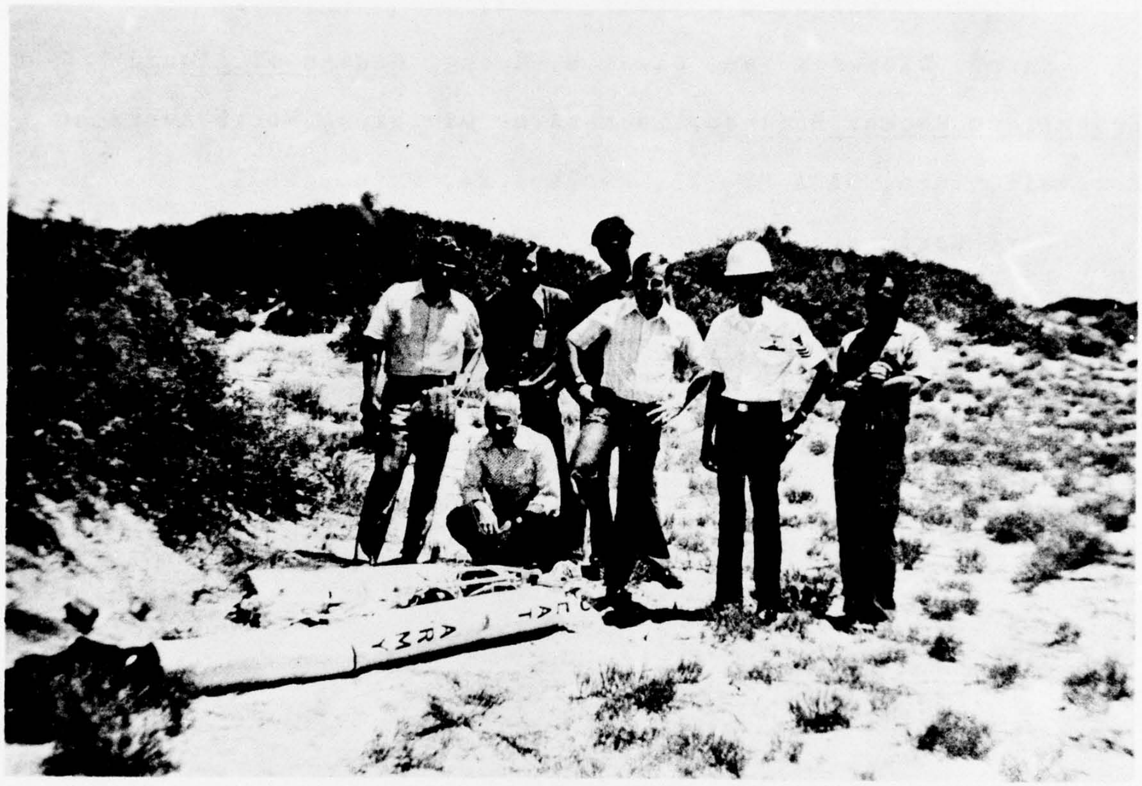

Figure 5j

Launch crew and the wreckage. From left to right, standing: Dr. A. A. Pouring, Art Goehring, GMM1 Whitby, Steve Brown, HM1 Ward, BM2 Busby. Kneeling: Emanuel Crisalli.

BIBLIOGRAPHY

Dommasch, Daniel O., et al, Airplane Aerodynamics, New York, Pitman Publishing Company, 1961.

Huzel, Dieter K. and David H. Huang, Design of Liquid Propellant Rocket Engines, Rocketdyne Division, North American Rockwell, Inc., NASA SP-125, Washington, D. C., 1971.

Kit, Boris and Douglas S. Evered, Rocket Propellant Handbook, New York, The MacMillan Company, 1960.

Sutton, George P., Rocket Propulsion Elements, John Wiley and Sons, Inc., New York, 1967.

FOOTNOTES

[1]Film Cooling - Injecting a percentage of the total fuel flow through rows of fine holes in the chamber and nozzle wall. The resulting film protects the exposed wall from excessive heat transfer.

[2]Regenerative cooling - The engine is cooled by pumping one of the propellants through a jacket surrounding the inner chamber wall and thereby removing heat.

[3]George P. Sutton, Rocket Propulsion Elements, New York, p. 172.

[4]Ibid, p. 172.

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13. ABSTRACT

The paper describes the design, construction, and testing of a liquid fueled rocket. Design calculations for the engine and propulsion systems were performed, checked, and an engine was constructed to those specifications. A static test stand was constructed and instrumented to evaluate engine performance during full duration static firings.

The engine and propulsion system proved successful and were then incorporated into the flight vehicle. The remainder of the paper describes the design, construction, and testing of all flight hardware including an on-board camera system and recovery system.

Two unsuccesseful attempts to launch the missile were made on 11 May 1976 at White Sands Missile Range in New Mexico. The reasons for the failure were corrected and the missile was successfully launched on 17 May 1976.

DD FORM 1 NOV 65 1473 (PAGE 1)

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DD FORM 1 NOV 65 1473 (BACK)

(PAGE 2)

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